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EVALUATION OF TAKEOFF AND LANDING PERFORMANCE OF COMMERCIAL STOL AIRPLANES

M. Calcara*

ABSTRACT. The basic requirements for commercial STOL airplanes leading to the use of high by-pass ratio turbofans and of very advanced high-lift systems are briefly recalled.

With the method developed herein, a rapid evaluation of takeoff and landing performance may be made, allowing an easy comparison of different configurations.

The method takes into account: safety requirements (speed maneuvering margin, critical engine failure at takeoff, landing field length factor), passenger comfort and pilot limitations due to "human factors" (maximum rate of descent near the ground and reaction times).

A numerical example illustrates the use of simple graphs, which are based only on the more important project parameters.

1. Introduction /113**

The success of commercial STOL airplanes depends on the solution of numerous problems connected with the engineering development of airplanes capable of high flying speeds (Mach 0.7 - 0.8), and also of safe operation on runways approximately 600 meters long, with a noise level lower than that of modern jet airliners. Furthermore, the direct operating costs have to be kept within limits acceptable to the users, even with the short flights and the consequent modest rates which are typical of STOL operations. Of particular importance is the behavior of the airplane under conditions of turbulence:

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**Numbers in the margin indicate the pagination in the original foreign text.

Unsatisfactory flight performance during storms would have a negative influence on the economy of operation, both because the cruising speed would have to be decreased, and because potential users would not be too interested in an airplane incapable of providing the same flying comfort as modern jets. As a consequence, high wing loads (no lower than 350 kgm^{-2}), and relatively moderate lengths will be needed.

The operators require a high flying speed also because it allows a greater flexibility of flight scheduling, making the airplane economical even for much longer flights.

It is because of these considerations that builders have shifted their emphasis from turboprop to turbofan engines.

Concerning the noise level, it is generally believed that a 95 PNdB level at a distance of 150 meters from the plane is the maximum value, at least for "city center" operations. This involves using engines with low specific thrust (defined as the ratio between thrust and total air displacement) on the order of 20 kg/kg sec^{-1}. Engines with low specific thrust show a faster decrease in thrust with speed and height; this, however, does not impair the cruising characteristics of STOL airplanes in comparison with CTOL, because of the STOL takeoff thrust requirements, which are approximately double those for CTOLs. As a consequence, currently engines with a by-pass ratio on the order of 10 are being developed.

The need for STOL operations with high wing loads involves maximum load coefficients higher than the ones which can be obtained with mechanical high-lift systems. This is especially true on landing, where passenger comfort sets the limits for the maximum allowable deceleration on the ground.

Many solutions have been suggested to achieve high maximum load coefficients. Those reported in Figure 1 are of particular interest, and represent the designs most actively pursued at present by the leading aircraft industries in Europe and the United States [1].

The use of lifting engines for STOL operations is a very promising solution, because it is simple; however, it requires two different types of engines. In this regard, the use of propulsion engines with orientable thrust (the type installed in the H. S. Harrier airplane) represents a more compact solution, even if not devoid of other shortcomings.

High-lift systems with internal air jets, including the "augmentor wing" type (extensively tested by De Havilland-Canada), involve the use of engines designed to satisfy the severe air requirements: On the average, the thrust due to spilled air is 35% of the total thrust.

High-lift systems with external air jet are compatible with high by-pass ratio engines, and therefore more convenient because the specific thrusts which can be obtained are as large as the amounts of air to be drawn in. Furthermore, they are devoid of the mechanical complications inherent in internal air jet /114
high-lift systems. However, the advantages of a higher simplicity are considerably reduced by the fact that the controllability of the airplane is decreased in case the critical engine fails.

None of the solutions described has enough advantages with respect to the others to warrant an immediate and unequivocal choice; furthermore, for each of them, many problems still have to be solved and many technical difficulties will have to be overcome.

It is, however, very interesting to compare the potential characteristics of each of the solutions mentioned, using the actual data available. If we limit ourselves to the takeoff and landing performance, we can see the advantage of having a calculation method leading to a rapid evaluation based on the following factors: safety requirements (speed maneuvering margin, failure

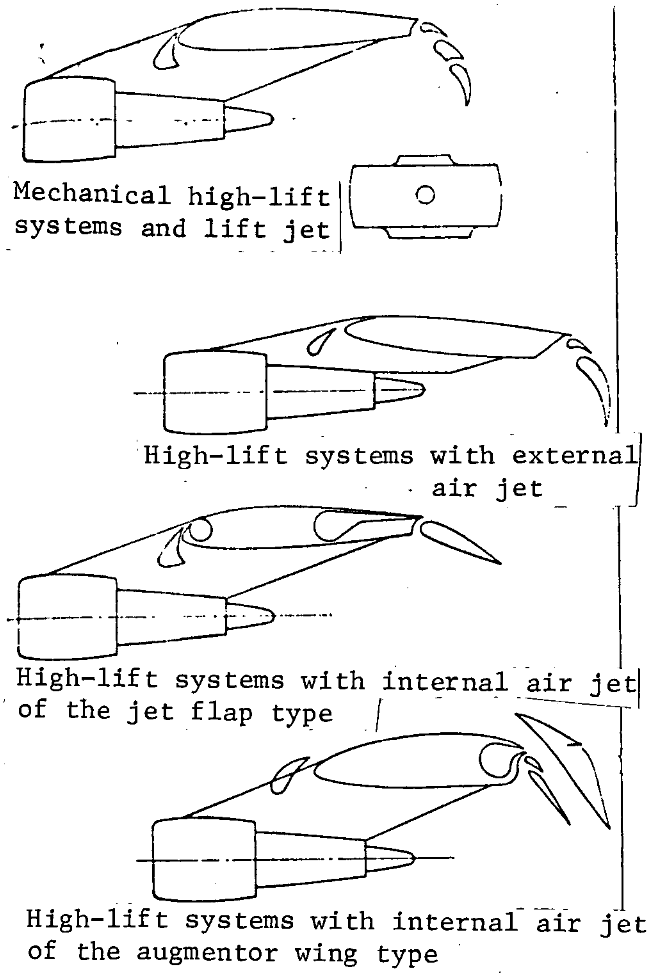

Figure 1. Advanced high-lift systems for STOL.

of critical engine at takeoff, landing field length factor), passenger comfort (maximum acceleration allowable on the ground), and "human factor" limitations (maximum rate of descent near the ground and reaction times).

Many procedures for a rapid evaluation of the takeoff and landing characteristics of STOL airplanes are available; however, they either provide only first approximations [2], or consider only normal conditions, omitting a study of the influence of critical engine failure or of reaction times [3].

2. Method of Calculation

The calculation procedure suggested applies to all STOL airplanes using jet engines, independently of the high-lift system used. It is based on the following hypotheses:

— attitude of the airplane with respect to the flight path and configuration with flaps in takeoff position for acceleration on the ground up to decision speed and subsequent deceleration in the case of postponed takeoff. Inclination of the jets of the lifting engines (if any) in order to obtain maximum possible acceleration or deceleration, respectively.

— attitude of the airplane in relation to the takeoff angle and configuration with flaps in takeoff position for acceleration on the ground from decision speed to takeoff speed. Inclination of the jets of the lifting engines (if any) so as to obtain maximum lift.

— decision speed not higher than 95% of the takeoff speed to implicitly allow for the time needed for the configuration change prior to takeoff.

— takeoff speed equal to speed maneuvering margin.

— two seconds time lag between occurrence of a critical engine failure and pilot noticing it.

— two seconds time for the pilot to "cut" the engine symmetrical to the one that failed and apply braking devices (wheel brakes, thrust inversion, and lowering of spoilers).

— angle of approach not above 7.5°, with descent speed not higher than 4 meters/second (considered by the pilots as a limit which can be reached with a reasonable safety margin).

— speed maneuvering margin same as impact speed.

— pilot reaction time to apply braking devices of two seconds after impact.

— attitude of the airplane with respect to flight path and configuration with lowered spoilers during the deceleration phase subsequent to landing.

— landing field length factor of 1/0.6, according to F.A.A. regulations for CTOL planes.

— limiting value for acceleration and deceleration of 0.5 g to assure acceptable passenger comfort level both during takeoff and landing.

— average engine thrust on takeoff assumed equal to 90% of the static thrust, with a 0.7 inversion thrust yield. Other numerical values introduced /115
in the calculations are reported in the table in Figure 2. One should note that, for the acceleration phase V_1 only, we did not consider the possible effect of air jets on the high-lift systems, because we believe it is negligible for this phase.

Phase	C_D	C_L	μ	
			Front wheel not braking	Front wheel braking
Acceleration at decision speed	0.08	0.70	0.025	
Failed takeoff deceleration	0.13	0	0.30	0.35
Landing: deceleration	0.20	0	0.30	0.35

Figure 2. Values of C_L, C_D, and μ assumed for the calculation of the takeoff and landing speed-up and slow-down operations.

The speed maneuvering margin has to be chosen so as to guarantee an adequate speed margin on stalling, and sufficient maneuvering capacity Δf_n:

$$\left(\frac{V_2}{V_s}\right)^2 = \frac{C_{L\max}}{C_{L2}} = 1 + \Delta f_n \qquad (1)$$

A speed maneuvering margin equal to 1.2 times the stalling speed with engines operating is currently considered sufficient for STOL. This corresponds to a maneuvering margin of 0.44 g; the value is reduced to 0.33 g in the case of one critical engine failure.

$$\Delta f_n = \begin{cases} 0.44 \text{ (all engines operating)} \\ 0.33 \text{ (failure of a critical engine)} \end{cases} \qquad (2)$$

Flight tests conducted with the Boeing 707 prototype equipped with control devices for the limiting level of the rear high-lift systems [4] showed that this criterion is valid as long as the speed margin ΔV is not lower than:

$$\Delta V = 5 \text{ m/sec} \qquad (3)$$

As a consequence, for values of $\frac{W/S}{\sigma\, C_{L\max}}$ below 39 kg m^{-2} (65 kg m^{-2} in case of engine failure), the following condition has to be satisfied:

$$\frac{C_{L\max}}{C_{L2}} = \left[\frac{\Delta V/4}{\sqrt{\frac{W/S}{\sigma\, C_{L\max}}}} + 1\right]^2 \qquad (4)$$

in place of (1).

The aerodynamic and propulsive operations are considered separately for all the phases on the ground, except for the increased attitude before takeoff (acceleration from decision speed to takeoff speed). For the latter phase and for the flight phases (speed maneuvering margin at takeoff and landing), we refer to the aerodynamic coefficients (C_L and C_D) which include the propulsive operations because, in this latter case, the mutual operation of wing and propulsion engines are relevant.

As a consequence, the following two acceleration expressions are used:

$$a = g\left[\frac{T}{W} - \mu - \frac{\sigma V^2}{16\, W/S}(C_D - \mu\, C_L)\right] \qquad (5)$$

$$a = -g\left\{\frac{\sigma V^2}{16\, W/S} C_{Da} + \mu\left[1 - \left(\frac{V}{V_a}\right)^2\right]\right\} \qquad (6)$$

which refer to the cases shown in Figures 3 and 4.

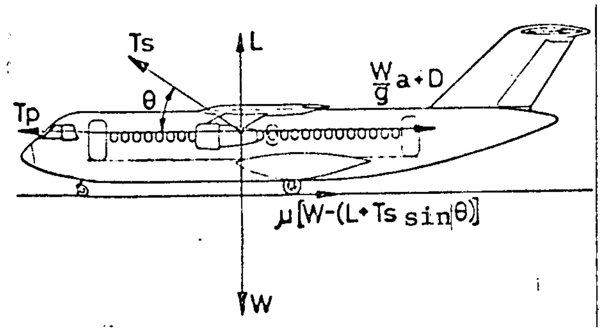

Figure 3. Takeoff and landing speed-up and slow-down operations. Airplane on three points of contact.

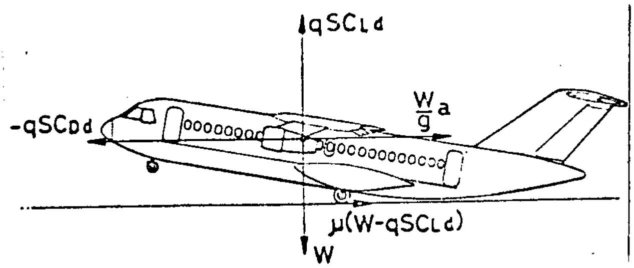

Figure 4. Takeoff and landing speed-up and slow-down operations. Airplane in takeoff attitude.

The ratio T/W appearing in the first expression is in general given by:

$$\frac{T}{W} = 0{,}9\,\tau \cdot \left[\frac{T_p}{W} + \frac{T_s}{W}(\cos\theta + \mu\sin\theta)\right] \tag{7}$$

During any phase of the run on the ground, one has, according to the energy theorem:

$$F ds = \frac{W}{2g}\, d(V^2).$$

Introducing the instantaneous acceleration ($a = \frac{F}{W/g}$) and integrating between the speed limits V' and V'', we obtain the following expression for the distance Δs covered by the airplane:

$$\Delta_s = \int_{V'}^{V''} \frac{1}{2a}\, d(V^2)$$

or, if a_m is the average acceleration:

$$\Delta_s = \frac{V''^2 - V'^2}{2\,a_m}. \tag{8}$$

As shown in (5) and (6), if we neglect the variation of T with speed, /117
the acceleration can be assumed to vary linearly with V^2. Therefore, a_m can be calculated using Expression (5) or (6) for the instantaneous value of the acceleration, if one assumes that the speed V* satisfies:

$$V^{*2} = 0{,}5\, V''^2 \left[1 + \left(\frac{V'}{V''}\right)^2\right]. \tag{9}$$

The takeoff distance S_d can be evaluated by determining the distance S_{d1} needed to reach the decision speed, the distance S_{d2} required to accelerate the airplane from the decision speed to takeoff speed, and the distance S_{d3} needed for the speed maneuvering margin.

$$s_d = s_{d1} + s_{d2} + s_{d3} \tag{10}$$

Speed V_1 can be determined with the assumption that (balanced runway):

$$s_{d2} + s_{d3} = s_{d4} + s_{d5} \tag{11}$$

where $S_{d4} + S_{d5}$ is the distance needed to stop the airplane in the case of failure during takeoff. Assumption (11) is not satisfied if $V_1 > 0.95\ V_d$.

The diagram in Figure 5 can be used to determine the distance:

$$s_{d1} = s'_{d1} + \Delta s_1 \tag{12}$$

where S'_{d1} is the distance needed to accelerate the airplane to the decision speed with all engines operating normally, and S_1 the correction due to critical engine failure, which, according to the hypothesis mentioned above, is assumed to occur at the speed V_0, reached two seconds before the decision speed V_1 is reached.

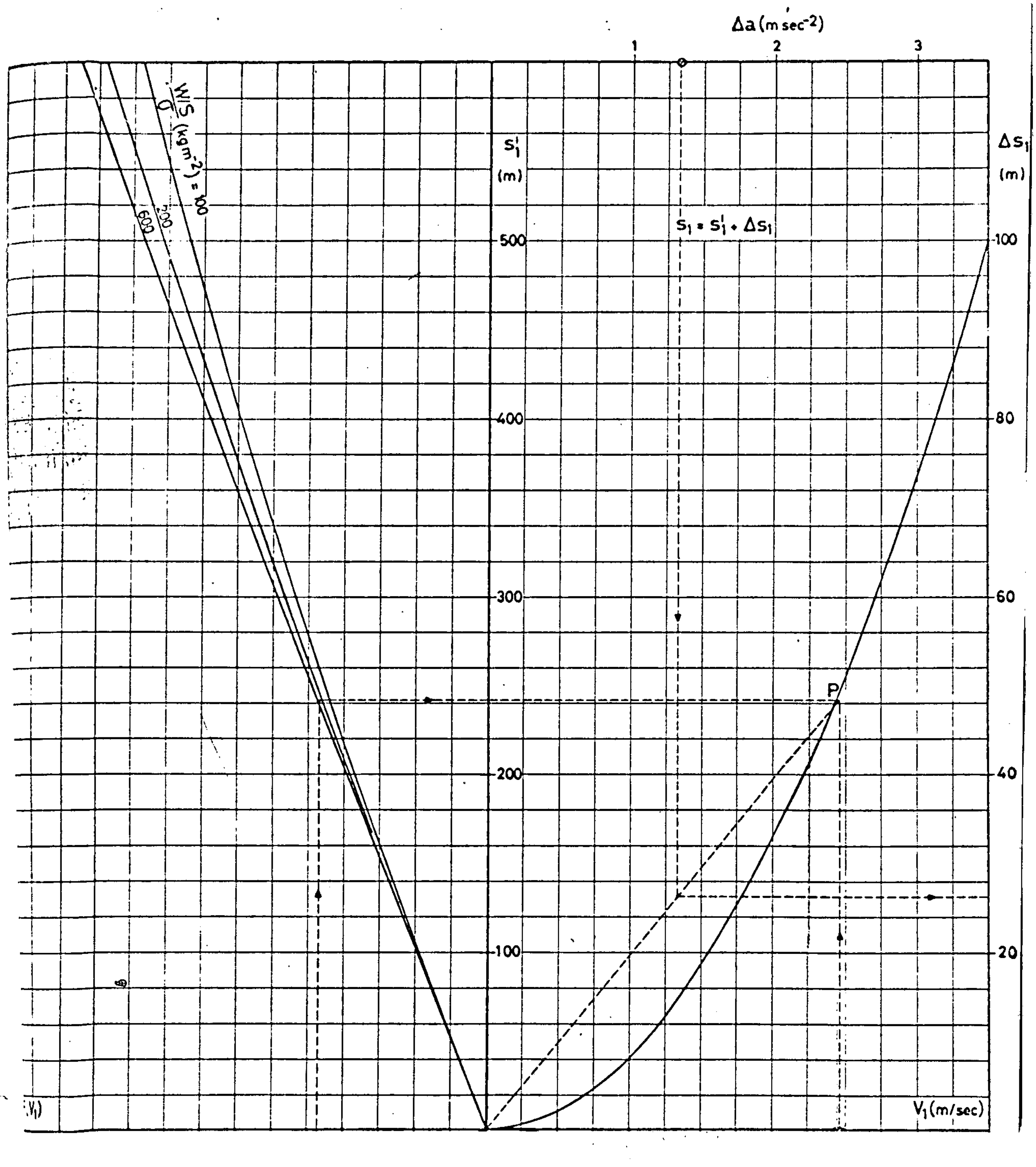

Figure 5. Takeoff distance: acceleration run at takeoff speed.

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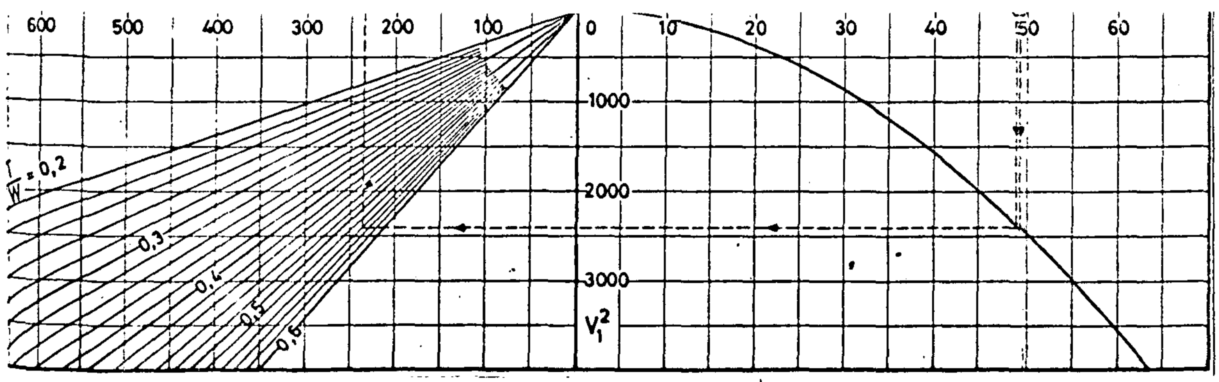

Figure 5 (concluded)

If a_0 is the average acceleration during the time interval Δt needed to go from V_0 to V_1 with all engines operating, and Δa the acceleration caused by the thrust of the critical engine only, (8) leads to the following expression for correction Δs_1:

$$\Delta_{s1} = \frac{V_1^2 - V_0^2}{2\,(a_0 - \Delta_a)} - \frac{V_1^2 - V_0^2}{2\,a_0} = \frac{\Delta a}{2\,a_0}\,\frac{V_1^2 - V_0^2}{a_0 - \Delta_a} \tag{13}$$

This expression is

$$\Delta_t = \frac{V_1 - V_0}{a_0 - \Delta_a} \tag{14}$$

and gives:

$$\Delta s_1 = \frac{\Delta a \Delta t}{2\,a_0}\,(V_1 + V_0) \tag{15}$$

With t = 2" second, we can assume $V_1 \simeq V_0$, and (15) gives the approximate expression:

$$\Delta s_1 = 2\,\frac{\Delta a}{a_0}\,V_1 \tag{16}$$

The distance S'_{d1} is given by (8) as:

$$s'_{d1} = \frac{V_1^2}{2\,a_1} \tag{17}$$

whereby, according to (5) and (9):

$$a_0 = g\left[\frac{T}{W} - \mu - \frac{\sigma\,V_1^2}{16\;W/S}\,(C_D - \mu C_L)\right] \tag{18}$$

$$a_1 = g\left[\frac{T}{W} - \mu - \frac{\sigma\,V_1^2}{32\;W/S}\,(C_D - \mu C_L)\right] \tag{19}$$

The first result is based on the same hypothesis ($V_1 \simeq V_0$), which makes (16) valid. If we assume:

$$F(V_1) = \frac{V_1^2}{2g\left(\frac{T}{W} - \mu\right)} \qquad (20)$$

(17) can also be written as follows, according to (19):

$$s'_{d1} = \frac{1}{\frac{1}{F(V_1)} - \frac{\sigma g}{16\, W/S}(C_D - \mu C_L)} \qquad (21)$$

The diagram in Figure 5 is based on Expressions (20) and (21), and leads to rapid determination of the distance S'_{d1}.

The quantity ΔS_1, on the other hand, can be obtained from:

$$\frac{V_1\, \Delta s_1}{4\, \Delta a} = \frac{V_1^2}{2\, a_0} = \frac{1}{\frac{1}{F(V_1)} - \frac{\sigma g}{8\, W/S}(C_D - \mu C_L)} \qquad (22)$$

which can be easily obtained from (16), (18), and (20). Comparing Expressions (21) and (22), we can assume that $V_1\, \Delta s_1 / 4\, \Delta a$ can be determined using the same diagram used to determine S'_{d1} as long as we refer to a wing load which is one-half of the actual one. As shown in the same diagram in Figure 5, the influence of the wing load is very small. We may, therefore, set

$$s'_{d1} = \frac{V_1\, \Delta s_1}{4\, \Delta a} \qquad (23)$$

According to the last expression, correction ΔS_1 can be read — on the S'_{d1} scale — as the ordinate of the line OP corresponding to the abscissa $4\ \Delta a$, read in the V_1 scale.

For reading convenience, we set up appropriate scales for ΔS_1 and proportional to those for S'_{d1} and V_1.

In the diagram the curve corresponding to a 0.5 g acceleration is also shown, which can be expressed, according to (8), as:

$$s'_{d1} = \frac{V_1^2}{g} \tag{24}$$

Points P lying in the space between axis V_1 and the curve of Equation (24) represent takeoff conditions with accelerations higher than 0.5 g, and therefore are not acceptable for passenger comfort.

In Figure 6 we show the diagram for determining the distance $S_{d2} + S_{d3}$. In particular, distance S_{d2} needed to accelerate the airplane from V_1 to V_d is given by (8) as:

$$S_{d2} = \frac{V_d^2 - V_1^2}{2\,a_2} \tag{25}$$

or

$$V_d^2 = \frac{16\ W/S}{\sigma\,C_{Ld}} \tag{26}$$

where the average acceleration a_2 is given by (6) and (9):

$$2\,\frac{a_2}{g} = -\left[1 + \left(\frac{V_1}{V_d}\right)^2\right]\frac{1}{E_d} - \left[1 - \left(\frac{V_1}{V_d}\right)^2\right]\mu \tag{27}$$

As a consequence, distance S_{d2} is obtained from

$$S_{d2} = -\frac{V_d^2}{\left[\frac{1}{E_d\varphi(V_1/V_d)} + \mu\right] g} \tag{28}$$

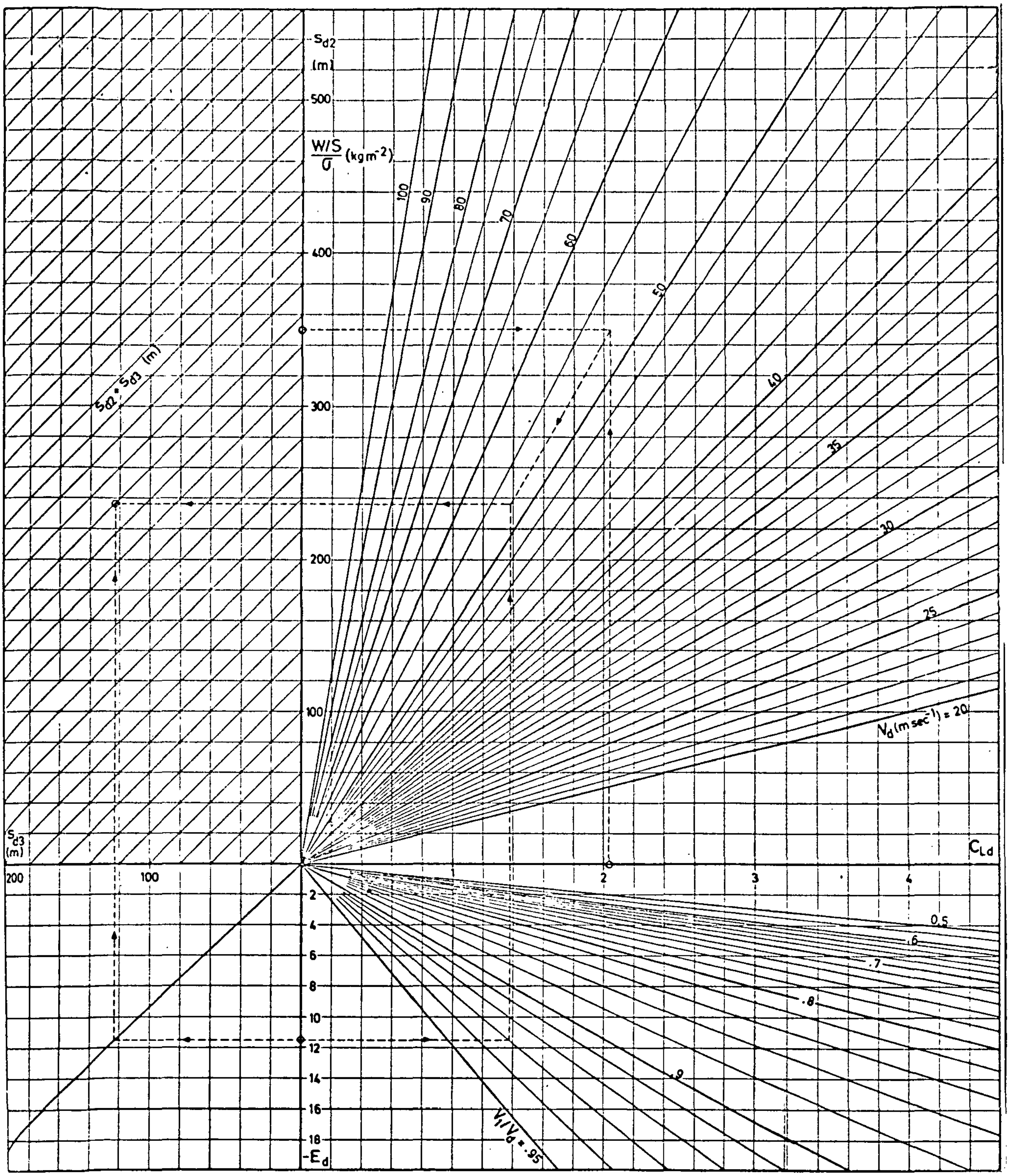

Figure 6. Takeoff distance: acceleration run from decision speed to takeoff speed and distance for speed maneuvering margin.

where /119

$$\varphi\left(\frac{V_1}{V_d}\right)=\frac{1-(V_1/V_d)^2}{1+(V_1/V_d)^2} \tag{29}$$

Comparing (26) and (28), we obtain:

$$\frac{C_{Ld}}{16}=-\frac{1}{g\left[\frac{1}{E_d\varphi(V_1/V_2)}+\mu\right]}$$

according to which, distance S_{d2} varies with takeoff speed in the same way as the parameter $\frac{W/S}{\sigma}$. Accordingly, one can use the same diagrams to evaluate both V_d and S_{d2}.

Finally, distance S_{d3} is determined from:

$$s_{d3}=-H\frac{C_{LD}}{C_{Dd}} \tag{30}$$

where H is the speed maneuvering margin (equal to 10.67 m according to F.A.A. regulations).

The diagram in Figure 6 was obtained from Expressions (26), (28), (29), and (30). For specific values of E_d, C_{Ld} and $\frac{W/S}{\sigma}$ — the takeoff speed V_d and the distance $S_{d2}+S_{d5}$ can be determined from it for various values of the ratio V_1/V_d.

Finally, the distance $S_{d4}+S_{d''}$ needed to stop after takeoff failure can be obtained as follows:

$$s_{d4}=V_1\,\Delta t+\frac{a_0-\Delta a}{2}\,\Delta t^2 \tag{31}$$

or, for t = 2 seconds, with a good approximation

$$s_{d4} = 2\, V_1 \tag{32}$$

$$s_{d5} = -\frac{V_1^2}{2\, a_5} \tag{33}$$

according to (5) and (9), if T_R is the symmetrical braking operation obtained by thrust inversion of the operating engines, one obtains:

$$a_5 = -g\left[\frac{T_R}{W} + \mu + \frac{\sigma\, V_1^2}{32\; W/S}\, C_D\right] \tag{34}$$

if

$$G(V_1) = \frac{V_1^2}{2\, g\left(\frac{T_R}{W} + \mu\right)} \tag{35}$$

(33) can also be written as:

$$s_{d5} = \frac{1}{\frac{1}{G\;(V_1)} + \frac{\sigma\, g}{16\;\, W/S}\, C_D} \tag{36}$$

The diagram in Figure 7 was obtained from Expressions (35) and (36), which are related to a determination of the braking distance; the total stopping distance $F_{d5} + S_{d4}$ can also be obtained from the diagram. A friction coefficient of 0.3 was assumed. This value is suitable for concrete runways, and airplanes equipped with brakes on the main landing gear. The diagram can be used also for μ values different from the one given:

$$\mu = 0{,}3 + \Delta\mu$$

as long as an "equivalent" value of the braking thrust is used

$$\left(\frac{T_R}{W}\right)_e = \frac{T_R}{W} + \Delta\mu$$

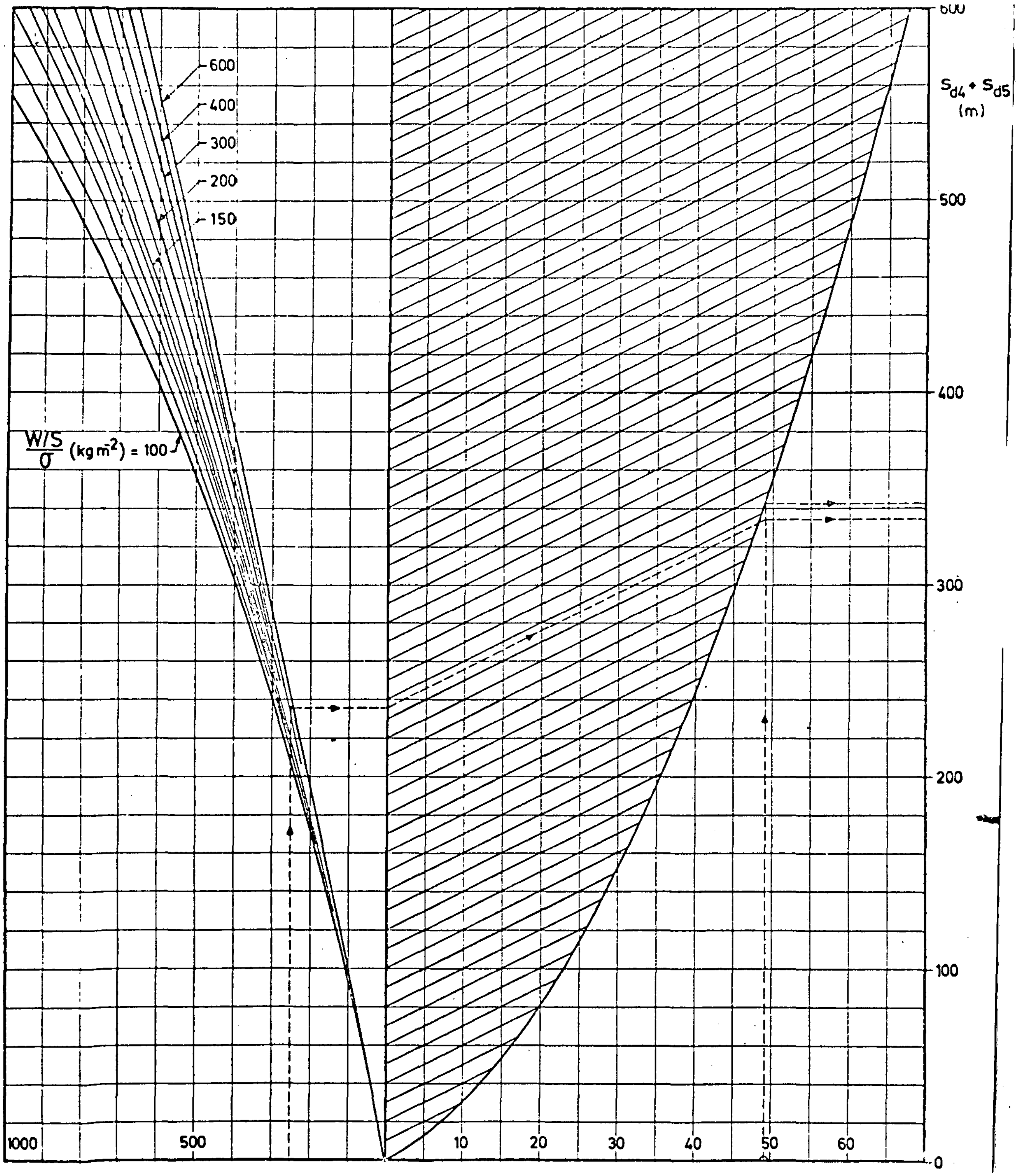

Figure 7. Takeoff distance: distance needed to stop after takeoff failure.

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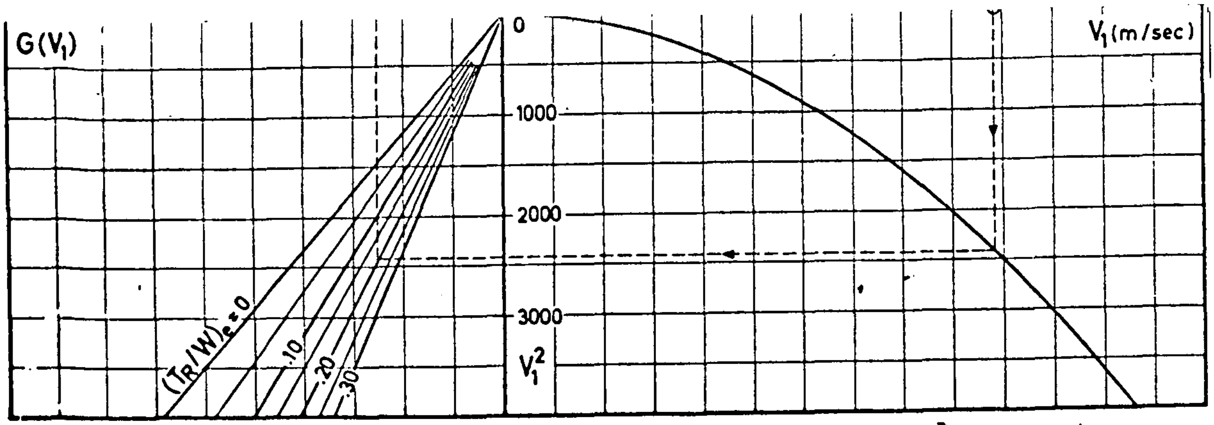
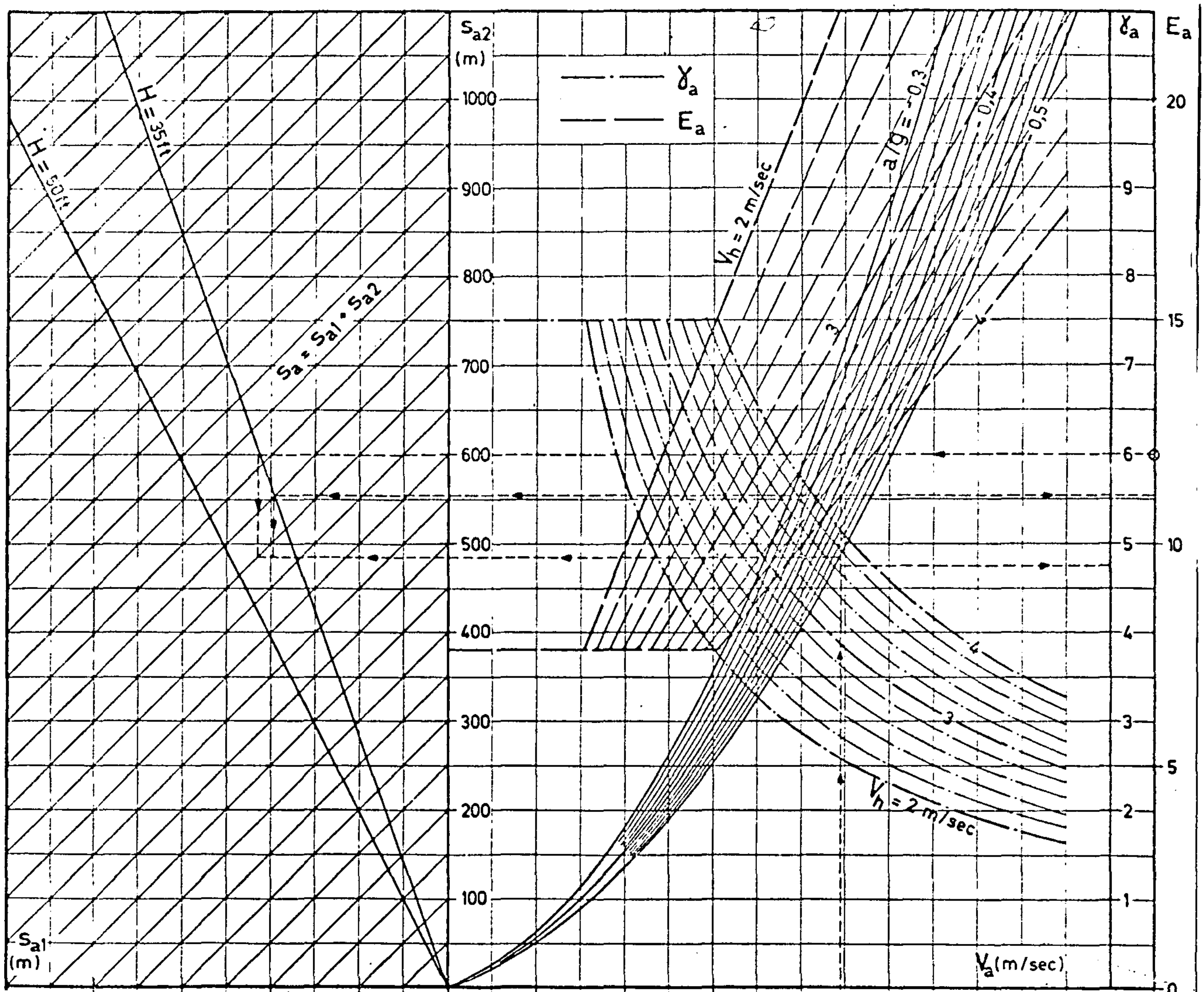


Figure 7 (concluded)



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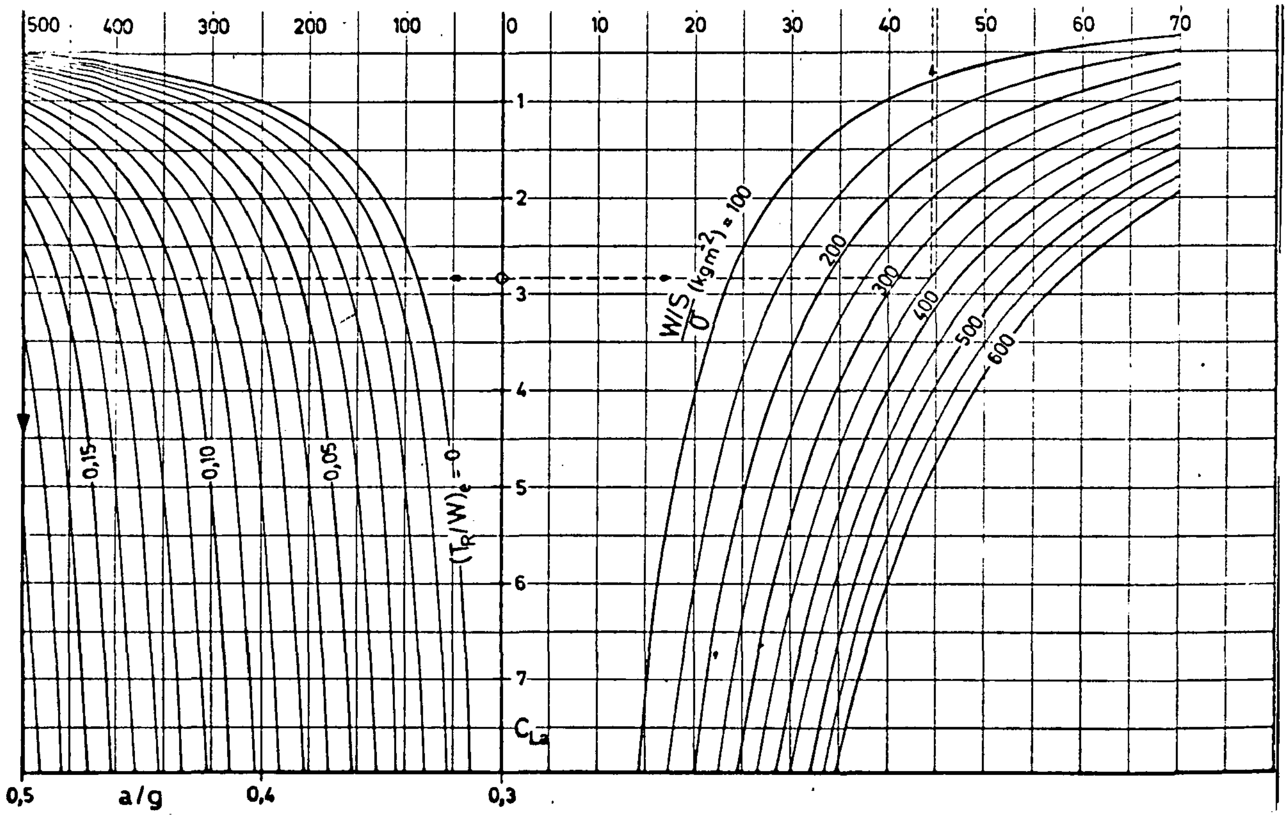

Figure 8. Landing distance.

In particular, when the front wheel is also involved in the braking action $\Delta\mu = 0.05$. The diagram also includes the 0.5 g deceleration limit considered to be the maximum acceptable for passenger comfort.

The diagram in Figure 8 provides a determination of the landing field length according to F.A.A. regulations (landing field length factor: 1/0.6):

$$s_a = s_{a1} + s_{a2} \tag{37}$$

S_{a1} is the distance corresponding to the speed maneuvering margin:

$$s_{a1} = \frac{HE_a}{0,6} \tag{38}$$

where

$$E_a = 1/\mathrm{tg}\ \gamma_a \tag{39}$$

with γ_a given by

$$\sin \gamma_a = V_h/V_a \tag{40}$$

for the limits:

$$V_h \leqslant 4\ m\ \mathrm{sec}^{-1}; \qquad \gamma_a \leqslant 7{,}5^\circ \tag{41}$$

The stopping distance S_{a2} is determined from

$$s_{a2} = \frac{1}{0,6}\left(2\ V_a + \frac{V_a^2}{2\ a}\right) \tag{42}$$

where the first term in parentheses represents the distance covered during the time delay needed to apply the braking devices (assumed to be two seconds), and the second term is the braking distance expressed as a function of the landing speed

$$V_a = 4\sqrt{\frac{W/S}{\sigma C_{La}}} \tag{43}$$

and the average deceleration, given according to (5), (9), and (43), by:

$$a = -g\left[\frac{T_R}{W} + \mu + \frac{C_D}{2 C_{La}}\right] \tag{44}$$

The nomogram in Figure 8 was obtained using Expressions (37) through (44), and assuming the Reference value μ = 0.3. It can be used for a rapid determination of the landing field length as a function of C_{La}, $\frac{W/S}{\sigma}$, $[T_R/W]_r$, H. /122

The takeoff and landing performance can be obtained from the diagrams in Figures 5, 6, 7, and 8, as follows, once parameters Δa, T/W, T_R/W, C_{L2} and C_{D2} are known.

The decrease in acceleration Δa due to critical engine failure — needed to determine S_{d1} — can be obtained from the diagram in Figure 9, which is a graphical representation of the functions:

$$(\Delta a)_p = 0{,}9\, g\,(1 - K'_p)\, \tau\, \frac{T_p}{W} \tag{45}$$

$$(\Delta a)_s = 0{,}9\, g\,(1 - K'_s)\, \tau\, \frac{T_s}{W}\,(\cos\theta + \mu \sin\theta) \tag{46}$$

concerning the failure of a propulsion engine or a lifting engine, respectively. Coefficient K' assumes the values 3/4, 5/6 and 7/8, respectively, for airplanes with 4, 6 or 8 engines. Comparing (7) with (45) and (46) we obtain:

$$\frac{T}{W} = \frac{1}{g}\,[(\Delta a)_p + (\Delta a)_s]_{K'=0} \tag{47}$$

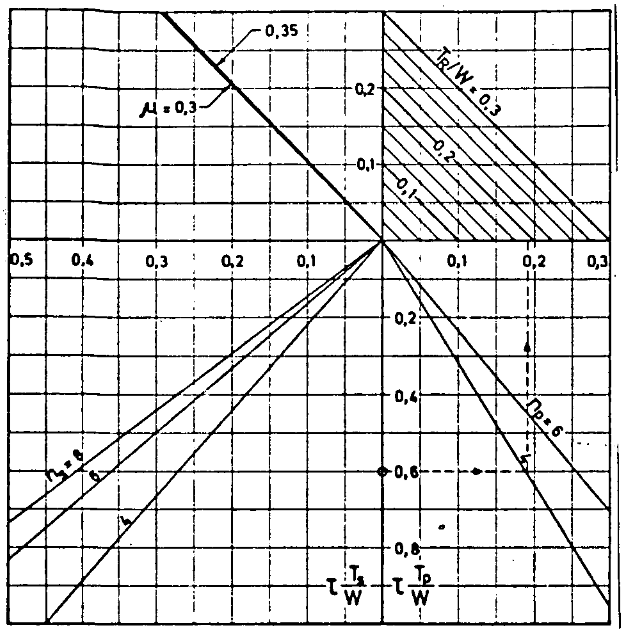

Figure 9. Diagram for the determination of Δa and T_R/W.

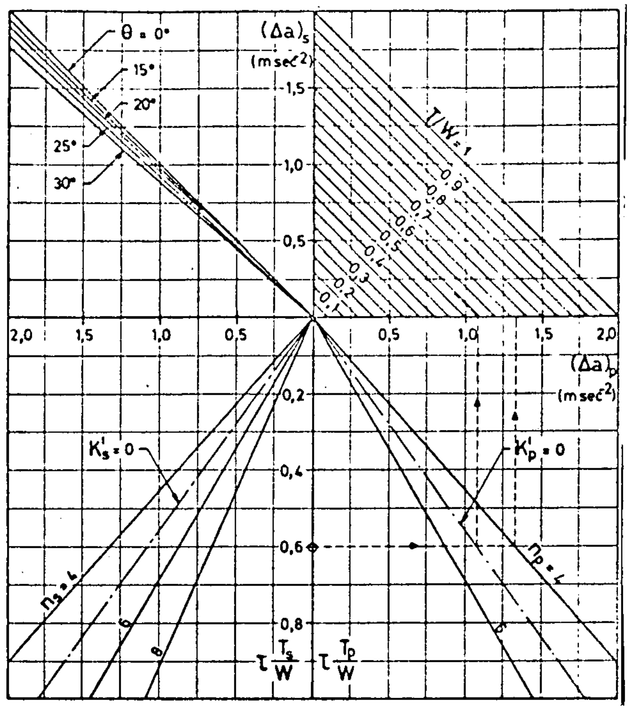

Figure 10. Diagram for determining P_R/W.

The diagram can also be used to determine T/W needed to obtain S_{d1}.

The diagram in Figure 10 can be used to obtain T_R/W, needed to determine the stopping distance $S_{d3} + S_{d4}$. The diagram was obtained from the expression:

$$\frac{T_R}{W} = 0{,}9\,\tau\left[\eta K''_p \frac{T_p}{W} + 0{,}9 K''_s \frac{T_s}{W}(\cos\theta + \mu \sin\theta)\right] \qquad (48)$$

where η is the thrust inversion efficiency, assumed equal to 0.7 and K" is a coefficient pertaining to the critical engine failure and the subsequent stopping of the symmetric engine. K" is equal to 1/2, 2/3, and 3/4, respectively, for airplanes with 4, 6, and 8 engines.

C_{L2} and E_2 can be determined from the following general expressions

$$C_{L2} = \frac{C_{L2'}}{1 - 0{,}9\,\tau\,K_s\,\dfrac{T_s}{W}} \tag{49}$$

and

$$E_2 = \frac{C_{L2}}{C_{D2} - (1 - \nu)\,C_{Tp}} \tag{50}$$

where:

$$C_{L2'} = \frac{C_{L\max}}{1 + \Delta f_n} \tag{51}$$

$$\frac{C_{Tp}}{C_{L2}} = 0{,}9\,\tau\,K_p\,\frac{T_p}{W} \tag{52}$$

$$\nu = \Delta T_p / T_p \tag{53}$$

where ΔT_p is the portion of propulsive thrust used for the air jets of the high-lift systems. One should note that coefficients C_{L2} and C_{D2} appearing in the expressions above are a function of the angle of the high-lift systems and of the air jet coefficient $C\mu$, which can be calculated from:

$$\frac{C\mu}{C_{L2}} = \nu\,\frac{C_{Tp}}{C_{L2}}\,\frac{C_{L2}}{C_{L2'}} \tag{54}$$

If this parameter is known, we can determine C_{L2} (as shown in Figure 11) and C_{D2}, which are needed to determine, through (49) and (50), parameters C_{L2} and E_2. In particular: /123

— for mechanical high-lift systems and lift engines:

$$\nu = 0$$

— for high-lift systems with internal air jet

$$\frac{T_s}{W} = 0$$

— for high-lift systems with external air jet

$$\nu = 1; \quad \frac{T_s}{W} = 0$$

The reduction coefficients K_S and K_p appearing in Expressions (49) and (52) refer to a possible critical engine failure. They are equal to 1 for all engines working, and, in case of failure, can be calculated from:

$$K_s = 1 - \frac{2}{n_s} = K''_s \tag{55}$$

$$K_p = 1 - \frac{1}{n_p} = K'_p \tag{56}$$

$$K_p = \left(1 + \frac{1}{0{,}7}\right)\left(\frac{1}{2} - \frac{1}{n_p}\right) \tag{57}$$

where n_p and n_S are the number of propulsion engines and lift engines, respectively.

Expression (56) is valid only for mechanical or internal air jet high-lift systems, because in these cases the maneuverability of the airplane is not compromised by a critical engine failure. For internal air jet high-lift systems, this is due to the interconnections of the air jet pipes.

Equation (57) applies to external air jet high-lift systems, and was derived from experimental results. According to these, the airplane can still be controlled, as long as the ratio between the propulsive thrust acting on the half wing with engine failure and the propulsive thrust acting on the other half wing does not fall below 7/10 [5].

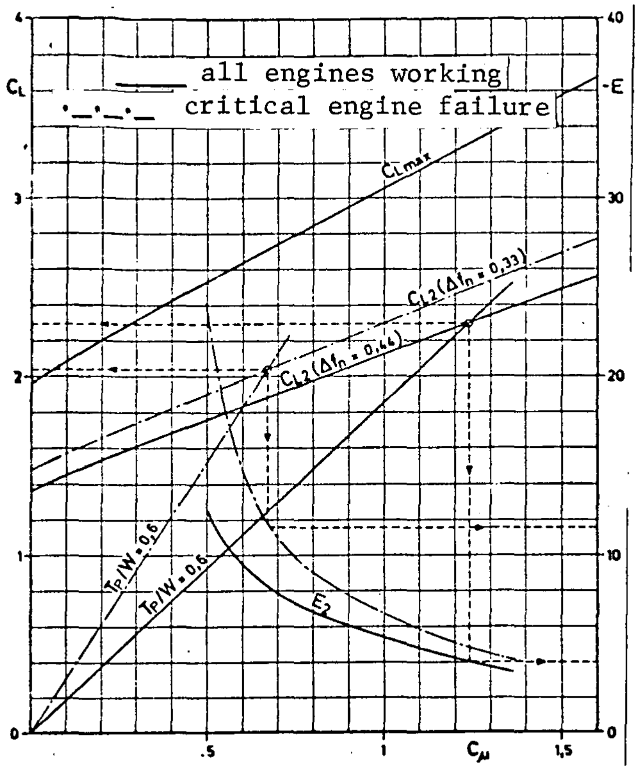

Figure 11. Determination of C_{L2}.

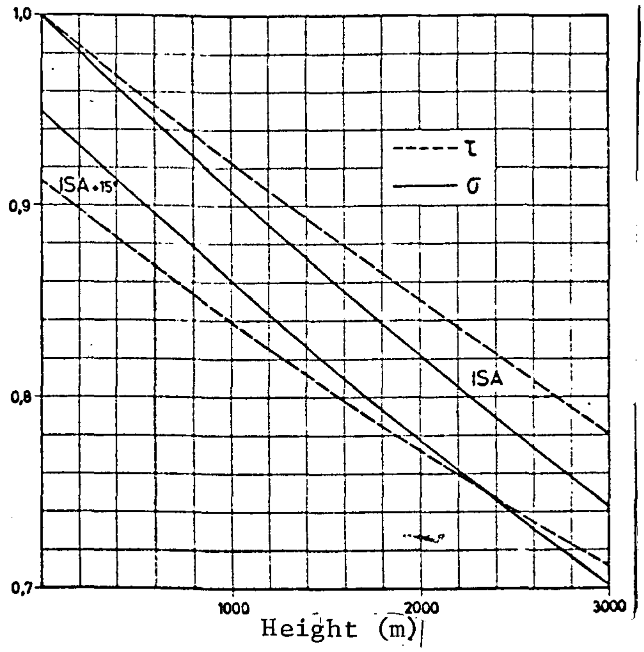

Figure 12. Diagram for the determination of σ and τ.

Expression (55) applies to lifting engines, and is based on the hypothesis that with critical engine failure the symmetrical engine will also have to be "cut".

The influence of local temperature and pressure conditions on takeoff and landing characteristics can be determined from the diagram in Figure 12, where the σ and τ values were plotted as a function of the height above sea level of the airport under ISA and ISA + 15° conditions.

3. Numerical Example

We want to determine the STOL performance under ISA conditions and at sea level ($\sigma = 1$, $\tau = 1$) for a commercial airplane equipped with external air jet high-lift systems, propelled by four turbofan engines, to which the following characteristic ratios would apply:

$$\frac{W}{S} = 350 \text{ kg m}^{-2} \qquad \frac{T_p}{W} = 0{,}6$$

The values of the load coefficients C_{L2} and the corresponding values of E_2 as a function of the air jet coefficient $C\mu$ for takeoff and landing configurations are known. The values of coefficients C_{L2}, for (51) and (2), are determined once the maximum load coefficients for the configuration studied /124

$$C_{L2} = \begin{cases} C_{Lmax}/1.44 \text{ (all engines working)} \\ C_{Lmax}/1.33 \text{ (critical engine failure)} \end{cases}$$

are known from Figure 11, which pertains to the takeoff configuration of the airplane studied.

Because $\nu = 1$ and T_S/W, Expressions (49) and (50) give respectively:

$$C_{L2} = C_{L2} \qquad E_2 = E_2$$

Consequently, since $n_p = 4$ (54), (52), and (57) give:

$$\frac{C\mu}{C_{L2}} = \frac{C_{Tp}}{C_{L2}} = \begin{cases} 0.9 \times 0.6 = 0.54 & \text{(all engines working)} \\ 0.54 \times 0.605 = 0.327 & \text{(critical engine failure)} \end{cases}$$

and therefore (Figure 11):

$$C_{Ld} = C_{L2} = \begin{cases} 2.29 & \text{(all engines working)} \\ 2.04 & \text{(critical engine failure)} \end{cases}$$

and correspondingly:

$$E_d = E_2 = \begin{matrix} -4.0 & \text{(all engines working)} \\ -11.5 & \text{(critical engine failure)} \end{matrix}$$

The diagram in Figure 6 gives, for $W/S = 350$ kg m^{-2} and $C_{Ld} = 2.04$, a value of 52.4 m/sec for the takeoff speed V_D. Furthermore, because:

$$E_d = -11.5 \text{ and } V_1 = 0.93, \; V_d = 48.7 \text{ m/sec}$$

we have:

$$S_{d2} + S_{d3} = 360 \text{ m}$$

On the other hand, the diagram in Figure 10, when $T_p/W = 0.6$, $T_S/W = 0$, $\tau = 1$, $n_p = 4$, gives a value of 0.19 for the parameter T_R/W, to which corresponds, for $\mu = 0.3$:

$$\left(\frac{T_R}{W}\right)_e = 0{,}19$$

This value entails a stopping distance $S_{d4} + S_{d5}$ equal to 335 m to counterbalance the kinetic energy of the airplane at a warm-up speed of 49 m/sec (Figure 7). Furthermore, the limiting condition of a 0.5 g acceleration requires a stopping distance slightly longer (343 m), but still in the same range as $S_{d2} + S_{d3}$. This confirms the value of the decision speed assumed above.

For $T_p/W = 0.6$, $\tau = 1$, $T_S/W = 0$, and $n_p = 4$, Figure 9 gives:

$$\frac{T}{W} = 0.54 \qquad (\Delta a)_p = 1.32$$

Consequently, the diagram in Figure 5 gives:

$$S'_{d1} = 242 \text{ m} \qquad \Delta S_1 = 26 \text{ m}$$

In the case of critical engine failure, the takeoff distance is therefore equal to:

$$S_d = 628 \text{ m}$$

For the case of a takeoff with no failure ($C_{Ld} = 2.29$ and $E_d = -4$), Figure 6 gives:

$$V_d = 49.5 \text{ m sec}^{-1},$$

a lower value than obtained above. Assuming $V_d = 52.5$ m sec^{-1} and, therefore, $E_d = -4.8$, the diagram in Figure 6, for $V_1 = 0.93\ V_d$, gives:

$$S_{d2} + S_{d3} = 150 \text{ m}$$

Consequently, the distance needed for normal takeoff becomes

$$S_d = 392 \text{ m}$$

This latter value, an increase of 15%, is lower than the one obtained previously in the case of critical engine failure. Therefore, according to F.A.A. regulations for CTOL airplanes, the takeoff field length is equal to 628 m in this particular case.

Following a procedure similar to the one shown in Figure 11, and for a landing configuration and critical engine failure ($\Delta f_n = 0.33$), we obtain an air jet coefficient value (for $T_p/W = 0.6$) of 0.94, so that:

$$C_{La} = C_{L2} = 2.84 \quad E_a = E_2 = 12$$

Using the diagram in Figure 8, we obtain

$$\left(\frac{T_R}{W}\right)_e = 0{,}164 < 0{,}19$$

and, therefore, $a/g = 0.5$.

Furthermore:

$$V_a = 44.5 \text{ m sec}^{-1}$$

and, for $H = 35$ ft, $V_h = 4 \text{ m sec}^{-1}$

$$S_a = 685 \text{ m}$$

This last value corresponds to an efficiency of 11.1, which is lower than the value for the configuration studied ($E_a = 12$). Consequently, the flight distance becomes longer, and the diagram in Figure 8 gives a landing field length of

$$S_a = 700 \text{ m}$$

The corresponding descent speed is $V_h = 3.7 \text{ m sec}^{-1}$, and the angle of approach is $\gamma_a = 4.75°$.

SYMBOL LIST

/125

a — airplane acceleration

C_D — resistance coefficient (= D/qS)

C_D — total resistance coefficient (propulsion operations included)

C_L — load coefficient (= L/qS)

C_L — total load coefficient (propulsion operations included)

C_{Lmax} — maximum load coefficient

C_{Tp} — propulsion thrust coefficient (= $0.9\tau\ K_p T_p/qS$)

$C\mu$ — air jet coefficient (= $\nu\ C_{Tp}$)

D — airplane resistance

E — aerodynamic efficiency (= C_L/C_D)

E — C_L/C_D

g — acceleration of gravity

H — maneuvering margin

K_p — reduction factor for critical propulsion engine failure (for $V > V_1$)

K_p' — reduction factor for critical propulsion engine failure (acceleration to V_1 phase)

K''_p — reduction factor for critical propulsion engine failure (deceleration phase on the ground)

K_S — reduction factor for critical lifting engine failure (for $V > V_1$)

K'_S — reduction factor for critical lifting engine failure (acceleration to phase V_1)

K''_S — reduction factor for critical lifting engine failure (deceleration phase on the ground)

L — airplane load

n_p — number of propulsion engines

n_S — number of lifting engines

q — dynamic pressure $(= \sigma V^2/16)$

S — wing area

S — distance covered by the airplane

S_a — landing distance

S_{a1} — speed maneuvering margin (on landing)

S_{a2} — stopping distance

S_d — takeoff distance

S_{d1} — distance needed to accelerate the airplane to V_1

S_{d2} — distance needed to accelerate the airplane from V_1 to V_d

S_{d3} — speed maneuvering margin (on takeoff)

$S_{d4} + S_{d5}$ — total stopping distance for takeoff failure

T — total thrust

T_p — total static thrust of propulsion engines

T_R — total braking thrust

$(T_R/W)_e$ — $T_R/W + \Delta\mu$

T_S — total static thrust of lifting engines

V — airplane speed

V_a — landing speed

V_d — takeoff speed on leaving the ground

V_h — descent speed near the ground

V_S — stalling speed (with engines working)

V_1 — warm-up speed

V_2 — maneuvering speed

W — airplane weight

γ_a — angle of approach

Δ_a — decrease in acceleration due to critical engine failure

Δf_n — maneuvering margin

ΔT_p — amount of propulsion thrust used for the high-lift systems air jets

ΔV — speed margin $(= V_2 - V_s)$

$\Delta\mu$ — $\mu - 0.3$

η — thrust inversion efficiency

θ — thrust inclincation from the horizontal of the lifting engines

μ — runway friction coefficient

ν — T_p/T_p

σ — relative density of air

τ — ratio between static propulsion thrust on takeoff under local pressure and temperature conditions and under ISA conditions at sea level

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